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# FLIGHT INVESTIGATION OF BOUNDARY-LAYER AND PROFILE-DRAG CHARACTERISTICS OF SMOOTH WING SECTIONS OF A P-47D AIRPLANE

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ADVANCE CONFIDENTIAL REPORT

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# FLIGHT INVESTIGATION OF BOUNDARY-LAYER AND PROFILE-DRAG CHARACTERISTICS OF SMOOTH WING SECTIONS OF A P-47D AIRPLANE

By John A. Zalovcik

## SUMMARY

A flight investigation was made of boundary-layer and profile-drag characteristics of smooth wing sections of a P-47D airplane. Measurements were made at three stations on the wing: boundary-layer measurements were made on the upper surface of the left wing in the slipstream at 25 percent semispan; pressure-distribution measurements were made on the upper surface of the left wing at 63 percent semispan; and wake surveys were made at 63 percent semispan of the right wing. The tests were made in straight flight and in turns over a range of conditions in which airplane lift coefficients from 0.15 to 0.68, Reynolds numbers from $7.7 \times 10^6$ to $19.7 \times 10^6$, and Mach numbers from 0.25 to 0.69 were obtained.

The results of the investigation indicated a minimum profile-drag coefficient of 0.0062 for the smooth section at 63 percent semispan. At the highest Mach number attained in the tests, the critical Mach number was exceeded by at least 0.04 with no evidence of compressibility shock losses appearing in the form of increased width of the wake or increased profile-drag coefficient. For flight conditions approaching the critical Mach number, variations in Mach number of as much as 0.17 appeared to have no effect on the profile-drag coefficient.

In the slipstream, transition occurred at least as far back as 20 percent chord on the upper surface at low lift coefficients.

## INTRODUCTION

In order to obtain a comparison of the profile-drag characteristics of wing sections of low-drag and older types under similar flight conditions, tests have been made of two P-47 airplanes: the P-47D airplane, having Republic S-3 sections and the XP-47F airplane, having sections that varied from an NACA 66-series section at the plane of symmetry to an NACA 67-series section at the tip. The investigation of the wing sections of the XP-47F airplane included tests to determine the profile drag of a wing section outside the slipstream and the position of transition on sections inside and outside the slipstream. The results of this investigation are presented in reference 1.

The tests with the P-47D airplane reported herein were generally similar in scope to the tests with the XP-47F airplane except that the tests with the P-47D were extended to considerably higher Mach numbers in order to obtain some information on compressibility effects at Mach numbers through the critical value. The tests were made in straight flight and steady turns at various normal accelerations over a range of indicated airspeeds from 155 to 330 miles per hour at altitudes of 12,000 and 24,000 feet.

## SYMBOLS

| | |
|---|---|
| $c$ | section chord |
| $x$ | distance along chord from leading edge |
| $s$ | distance along surface from leading edge |
| $d$ | deflection of curvature gage |
| $y$ | distance above surface, position in wake |
| $H_o$ | free-stream total pressure |
| $H_y$ | total pressure in boundary layer |

| | |
|---|---|
| $\Delta H$ | loss of total pressure in wake |
| $p_o$ | free-stream static pressure |
| $p$ | local static pressure |
| $q_{c_o}$ | free-stream impact pressure $(H_o - p_o)$ |
| $q_o$ | free-stream dynamic pressure $\left(\frac{1}{2}\rho_o V^2\right)$ |
| $T_y$ | absolute temperature in boundary layer |
| $T_\delta$ | absolute temperature just outside boundary layer |
| $u$ | velocity in boundary layer |
| $u_1$ | velocity in boundary layer near surface |
| $U$ | velocity just outside boundary layer |
| $P$ | pressure coefficient $\left(\frac{p - p_o}{q_o}\right)$ |
| $P_{cr}$ | critical pressure coefficient, corresponding to local velocity of sound |
| $C_L$ | airplane lift coefficient |
| $c_{d_o}$ | section profile-drag coefficient |
| $\delta_a$ | aileron deflection, negative for up deflection |
| $V_c$ | calibrated airspeed (airspeed related to differential pressure by accepted standard adiabatic formula used in calibration of differential-pressure indicators and equal to true airspeed for standard sea-level conditions) |
| $V$ | true airspeed |
| $R$ | Reynolds number |
| $M_o$ | free-stream Mach number |
| $M_y$ | Mach number in boundary layer |
| $M_\delta$ | Mach number just outside boundary layer |

| | |
|---|---|
| $M_{cr}$ | critical Mach number |
| $g$ | acceleration of gravity |
| $\rho_o$ | free-stream density |

Subscripts:

| | |
|---|---|
| R | right |
| L | left |

## APPARATUS

The P-47D airplane is a low-wing, single-engine monoplane with a Pratt & Whitney R-2800-21 engine and a four-blade Curtiss electric propeller (fig. 1). The airplane has a gross weight of about 12,000 pounds, a wing span of 41 feet, and a wing area of 300 square feet. The wing incorporates Republic S-3 airfoil sections, which have pressure-distribution characteristics similar to those of the NACA 230-series sections.

Three wing sections were tested (fig. 1): one on the right wing and one on the left wing located 63 percent semispan from the plane of symmetry, or about 2 feet outboard of the flap (section with aileron); and one on the left wing located 25 percent semispan from the plane of symmetry, or about 1 foot within the edge of the propeller disk. Each of the outboard sections had a chord of 7.17 feet and a maximum thickness of 11 percent chord. The inboard section in the slipstream had a chord of 8.73 feet and a maximum thickness of 14.6 percent chord. A photograph of the test section on the right wing is shown as figure 2.

The upper surfaces of the sections on the left wing and the upper and lower surfaces of the section on the right wing were faired by filling with glazing putty and then sanding smooth to reduce the surface waviness. The surfaces were then sprayed with several coats of white lacquer-based paint for a protective coating and sanded lightly in a chordwise direction with No. 320 carborundum paper. An indication of surface waviness was obtained by means of a curvature gage (fig. 3) with legs spaced 4 percent of the wing section chord. The waviness condition of the faired surfaces is indicated in figure 4 by the plot of the waviness index d/c against s/c.

Boundary-layer racks, each consisting of one static-pressure tube and either one or five total-pressure tubes (fig. 5), were used to determine boundary-layer characteristics. The tubes were made of $\frac{1}{8}$-inch brass tubing with a $\frac{1}{32}$-inch wall thickness. The upstream end of the total-pressure tube was filed and flattened so as to leave an opening 0.003 inch deep and $\frac{1}{8}$ inch wide and to have a 0.003-inch wall thickness. The static-pressure tube had six orifices 0.02 inch in diameter equally spaced around the periphery at $1\frac{1}{4}$ inches downstream from the hemispherical end. Each total-pressure tube of a rack was connected to an NACA recording multiple manometer and referenced to the static pressure obtained from the static-pressure tube set about 1/4 inch from the surface. With this arrangement, the impact pressure was measured at various distances above the surface when the six-tube rack was used and near the surface when the two-tube rack was used. The static pressure measured by the static-pressure tube was referenced to the static pressure obtained by means of an airspeed head mounted on a boom 1 chord ahead of the leading edge of the right wing tip (fig. 1).

Surveys of the wake of the right wing section were made by means of the rake shown in figure 6 mounted 19 percent chord behind the trailing edge. The rake consisted of 24 total-pressure tubes spaced 0.3 inch and 5 static-pressure tubes spaced equally across the rake. The total-pressure tubes were connected to an NACA recording multiple manometer and referenced to free-stream total pressure in order that the total-pressure loss at each point in the wake could be obtained. The static pressure in the wake was measured with the three central static-pressure tubes, each of which was connected to the manometer, and referenced to the static pressure measured by means of the airspeed head on the boom at the right wing tip. Wool tufts were located on the upper surface near the trailing-edge region about 2 feet on each side of the center line of the section at 63 percent semispan to determine whether any cross flow existed that would invalidate the wake surveys.

All pressures, aileron positions, and normal accelerations were measured by NACA recording instruments. An indicating accelerometer was provided for the pilot.

## METHOD

In order to obtain free-stream static pressure, corrections determined from an airspeed calibration were made to the static pressure measured by the airspeed head mounted on the boom ahead of the right wing tip. These corrections were applied to all measurements for which reference to free-stream static pressure was required.

Boundary-layer velocity profiles were determined from the boundary-layer measurements by use of the compressible-flow relation

$$\frac{u}{U} = \left[\frac{\left(H_y/p\right)^{2/7} - 1}{\left(H_o/p\right)^{2/7} - 1}\right]^{1/2} \left(\frac{T_y}{T_\delta}\right)^{1/2}$$

$$= \frac{M_y}{M_\delta}\left(\frac{T_y}{T_\delta}\right)^{1/2}$$

or, to a first-order approximation,

$$\frac{u}{U} = \frac{M_y}{M_\delta}$$

The airplane lift coefficient at which transition occurred at a given chordwise position was determined from a plot of the ratio $u_1/U$ against airplane lift coefficient. The lift coefficient corresponding to transition was chosen at the elbow of the curve as the ratio $u_1/U$ suddenly increased from its laminar level to its turbulent level.

The profile-drag coefficients were determined by the integrating method of reference 2; that is, the total-pressure loss was integrated across the wake and then multiplied by factors depending on free-stream impact pressure, maximum total-pressure loss, static pressure in the wake, and flight Mach number.

## TESTS

Surveys of the wake of the smooth right wing section were made first in straight flight with level-flight power and with the airplane engine throttled and then in turns in order to cover a wide range of flight conditions; that is, airplane lift coefficients, Reynolds numbers, and Mach numbers. During the first flight in turns, the filler used to fair the wing surface cracked at the leading edge of the ammunition-compartment door (at 11.5 percent chord). Since the crack could not be kept smooth and the surface unbroken in subsequent flights, the wake surveys were discontinued.

Boundary-layer measurements were made both with the two-tube and the six-tube boundary-layer racks on the upper surface of the inboard section behind the propeller on the left wing. Measurements of static pressure and of impact pressure next to the surface for the determination of transition were made with two-tube racks at 5, 10, 15, 20, and 25 percent chord. Measurements of velocity distribution through the boundary layer were made with the six-tube racks at 15 and 20 percent chord.

Transition measurements on the upper surface of the outboard section on the left wing were not feasible because of the spanwise crack at the leading edge of the ammunition-compartment door at 11.5 percent chord. Static pressures, however, were measured with the static-pressure tubes of the boundary-layer racks at 10, 15, 20, 25, and 30 percent chord on the upper surface of this wing.

The tests were made in straight flight (level flight and shallow dives) at altitudes of 12,000 and 24,000 feet over a range of indicated airspeeds from 155 to 330 miles per hour. The airplane lift coefficients obtained in these tests ranged from 0.15 to 0.63; the Reynolds number, from $7.7 \times 10^6$ to $19.2 \times 10^6$; and the Mach number, from 0.25 to 0.69. Tests were also made in turns at an altitude of 12,000 feet at indicated airspeeds from 256 to 360 miles per hour and at normal accelerations from $1\frac{1}{2}g$ to $4\frac{1}{2}g$. The airplane lift coefficients in the turns ranged from 0.21 to 0.56; Reynolds number, from $14.2 \times 10^6$ to $19.7 \times 10^6$; and Mach number, from 0.44 to 0.61.

## RESULTS AND DISCUSSION

Pressure distribution and critical Mach number.- Some representative static-pressure distributions over part of the upper surface of the left wing sections at 25 and 63 percent semispan are shown in figure 7. The critical Mach numbers of the two wing sections, as determined by the von Kármán method (reference 3) from pressure-distribution measurements at subcritical speeds, are plotted in figure 8 against $\frac{C_L\sqrt{1 - M_0^2}}{\sqrt{1 - M_{cr}^2}}$, which represents the lift coefficient that would be obtained if the Mach number were increased from $M_0$ to $M_{cr}$ at the angle of attack corresponding to $C_L$. The flight Mach number and the deflection of the left aileron are plotted above the curves of critical Mach number.

For the section at 63 percent semispan, the critical Mach number varied approximately linearly from 0.66 at a lift coefficient of 0.10 to 0.54 at a lift coefficient of 0.80; for the section at 25 percent semispan, the variation of critical Mach number over the same range of lift coefficients was from 0.63 to 0.49. Although the evaluation of critical Mach number involved extrapolation by the von Kármán method of static-pressure data obtained at flight Mach numbers ranging from 0.02 to 0.30 below the critical value, the results were in good agreement for the entire range of the extrapolation. The extent of the extrapolation at various lift coefficients may be determined by comparing the flight Mach numbers at which the pressure-distribution measurements were made with the critical Mach numbers. (See fig. 8.)

According to the results presented in reference 4, the critical Mach numbers as determined from measurements with static-pressure tubes similar to those used in the present investigation may be as much as 0.01 higher than would be obtained from measurements with orifices flush with the wing surface.

It should be noted that, since the left aileron was deflected upward from 1.5° to 3.6° during the tests (fig. 8), the critical Mach numbers at 63 percent semispan may be somewhat higher than the critical Mach numbers that would be obtained with the aileron neutral. An indication of the magnitude of this effect is given in reference 5, which presents the results of tests of a model of a wing section with aileron on a P-47B-3 airplane. (The wing sections of this airplane are similar to those of a P-47D airplane.) The results in reference 5 showed that, at a constant angle of attack in the range of the flight tests, the critical Mach number was higher by about 0.015 with the aileron deflected upward 2° than with the aileron neutral.

Boundary-layer characteristics in slipstream.- The method of determining the airplane lift coefficient, section Reynolds number, and flight Mach number corresponding to transition from measurements with a boundary-layer rack in a given position on the wing surface is illustrated in figure 9 for a rack at 15 percent chord on the upper surface in the slipstream (at 25 percent semispan). The broken lines in this figure indicate the conditions for transition.

The results of the boundary-layer measurements indicated that, at low lift coefficients, laminar flow was obtained at least as far back as 20 percent chord on the upper surface, which is about as far back as may be expected on a similar wing section outside the propeller slipstream. Laminar flow at 20 percent chord is illustrated by typical velocity profiles in figure 10. The lift coefficients, Reynolds numbers, and Mach numbers at which transition was obtained at 10, 15, and 20 percent chord are given in figure 11. At lift coefficients and Reynolds numbers less than those indicated by the curves for 15 and 20 percent chord in figure 11, the flow was laminar at those chordwise positions. Although transition measurements were also made at 5 and 25 percent chord, these data were not presented, inasmuch as the flow was always laminar at 5 percent chord and always turbulent at 25 percent chord.

Profile drag of wing section outside slipstream.- During all the tests the wool tufts on the upper surface near 63 percent semispan of the right wing were directed straight back and thereby indicated that the wake surveys were not influenced by cross flow.

The profile-drag coefficients of the smooth section on the right wing are presented in figure 12 for straight flight and in figure 13 for turns. Flight Mach number, critical Mach number, Reynolds number, calibrated airspeed, and deflection of the right aileron are plotted above the profile-drag curves. The critical Mach number shown in figures 12 and 13 is that for the left wing section. Inasmuch as the right aileron was down (figs. 12 and 13) when the left aileron was up (fig. 8), the critical Mach number for the right wing section has been estimated on the basis of the results of reference 5 to be of the order of 0.02 lower than the critical Mach number of the left wing section. Some representative wake profiles obtained in straight flight are shown in figure 14.

In straight flight, the profile-drag coefficient varied from 0.0075 at a lift coefficient of 0.68 to 0.0062 at a lift coefficient of 0.15 (fig. 12). The minimum profile-drag coefficient was 0.0062. Within the accuracy of the measurements, changing from level-flight power to glides with engines throttled appeared to have no effect on the profile-drag coefficient.

The interpretation of the results of the profile-drag measurements in turns (fig. 13) is complicated by the fact that a crack developed at the leading edge of the ammunition-compartment door (at 11.5 percent chord) some time during the flight in which these measurements were made. The tendency toward lower profile-drag coefficients for the first series of turns than for the other series indicated that the crack may have developed after the first series of turns. At lift coefficients greater than 0.40, the profile-drag coefficients in turns agreed with those obtained in straight flight and thereby indicated that transition was probably forward of 11.5 percent chord at these high lift coefficients; at lift coefficients less than 0.40 the profile-drag coefficients for the second and third series of turns were somewhat higher than those obtained in straight flight. The minimum profile-drag coefficient for the second and third series of turns was 0.0066.

For flight conditions approaching the critical Mach number, a variation in Mach number as large as 0.17 (fig. 12) with a relatively small variation in Reynolds number appeared to have no effect on profile-drag coefficient. A similar result was obtained in the tests reported in reference 6 on the same wing section with transition fixed near the leading edge for smoothed and moderately roughened surfaces. A comparison of figures 12 and 13 shows that, at a lift coefficient of 0.47, the same value of profile-drag coefficient (within the experimental error) was obtained at Mach numbers varying from 0.30 to 0.59; in this case, however, the variation in Reynolds number was large ($10 \times 10^6$ to $19 \times 10^6$) and therefore may have had an effect on the results.

At the highest Mach number attained in the tests (0.69), the critical Mach number was exceeded by at least 0.04 (fig. 12) with no evidence of compressibility shock losses appearing in the form of increased width of the wake and increased profile-drag coefficient. This result appears to indicate either that irrotational flow without shock existed to some extent at supercritical speeds, as suggested in references 3, 7, and 8, or that the effect of compression shock was of insufficient magnitude to be measurable by present apparatus for a small range of Mach numbers above the critical value. Mild compression shocks have been indicated by Schlieren photographs obtained in wind tunnels of NACA 230-series airfoils. These photographs show that, upon attainment of local velocity of sound, shock first appears as a series of small shock waves and builds up to a well-established shock front as the Mach number is further increased.

## CONCLUSIONS

The flight investigation of boundary-layer and profile-drag characteristics of wing sections of a P-47D airplane that were specially finished to give aerodynamically smooth surfaces having waviness of small magnitude indicated the following results:

1. Boundary-layer transition at least as far back as 20 percent chord was obtained on the upper surface of a section in the slipstream at low lift coefficients.

2. In straight flight (level flight and shallow dives) the profile-drag coefficient of a section outside the slipstream varied from 0.0062 at a lift coefficient of 0.15 to 0.0075 at a lift coefficient of 0.63. The minimum profile-drag coefficient was 0.0062.

3. At the highest Mach number attained in the tests, the critical Mach number was exceeded by at least 0.04

with no evidence of compressibility shock losses appearing in the form of increased width of the wake or increased profile-drag coefficient.

4. For flight conditions approaching the critical Mach number, variations in Mach number as large as 0.17 appeared to have no effect on profile-drag coefficient.

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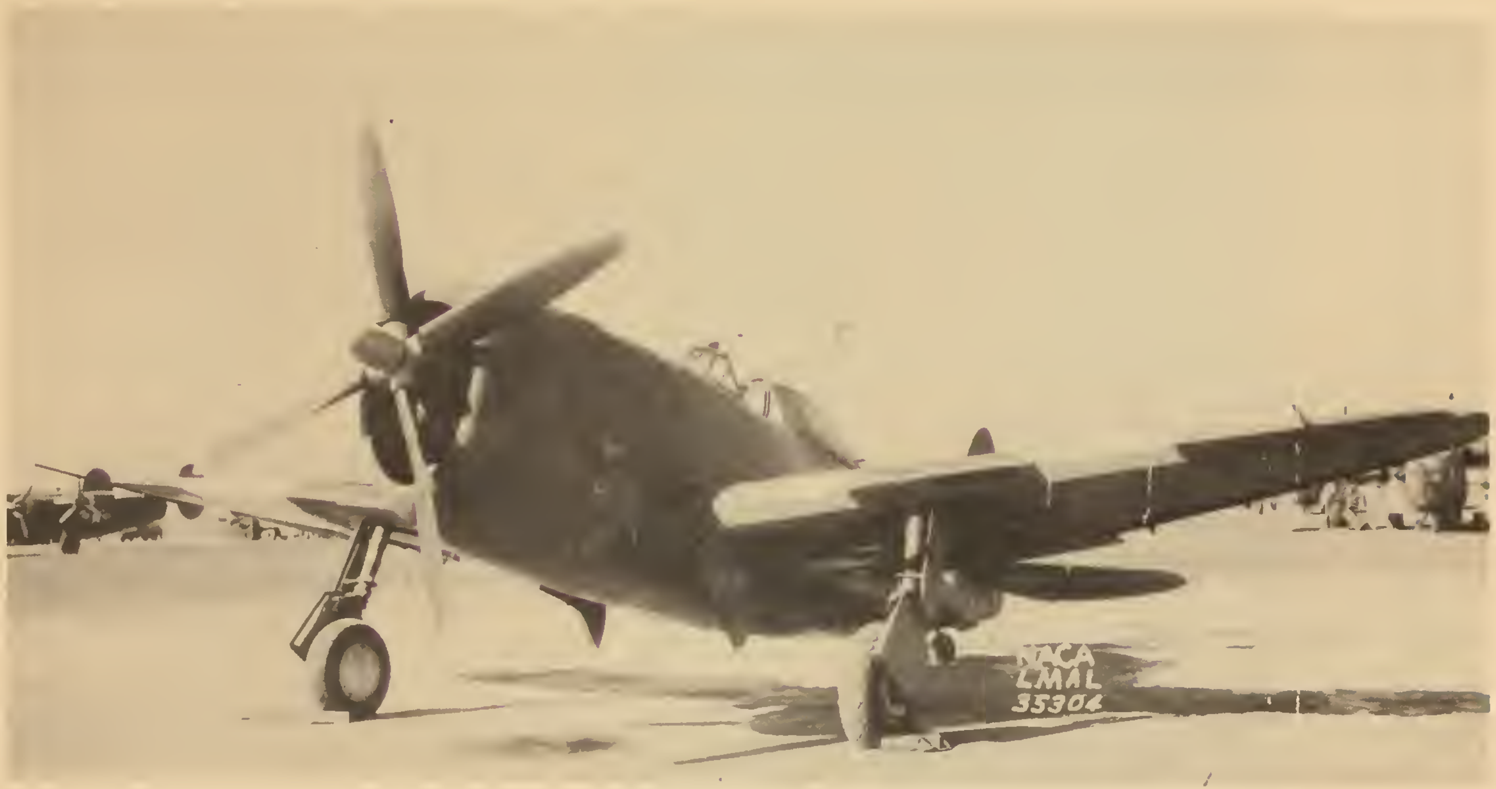

Figure 1.- Republic P-47D airplane with test panels refinished in white on right and left wings.

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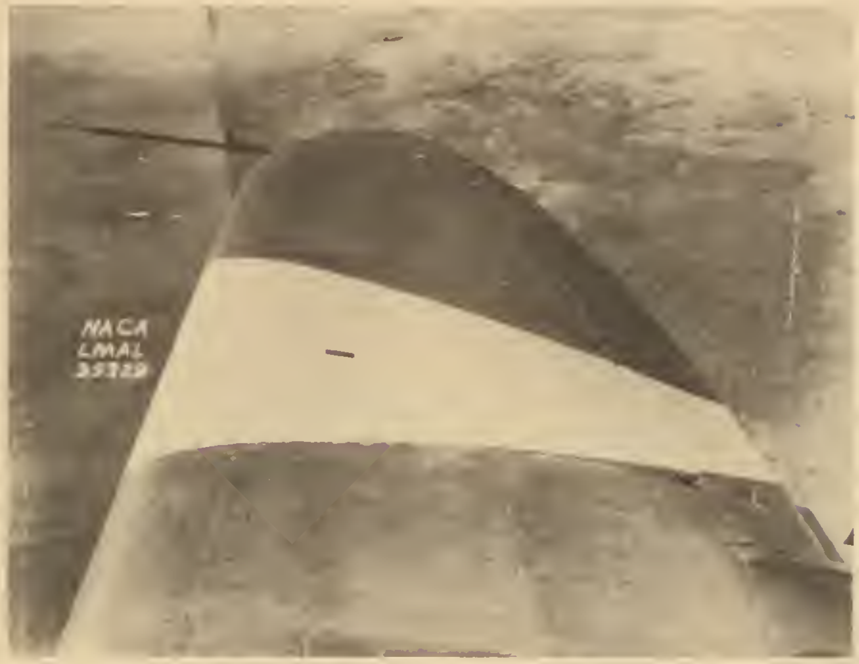

Figure 2.- Smooth test panel at 63 percent semispan on right wing of P-47D airplane.

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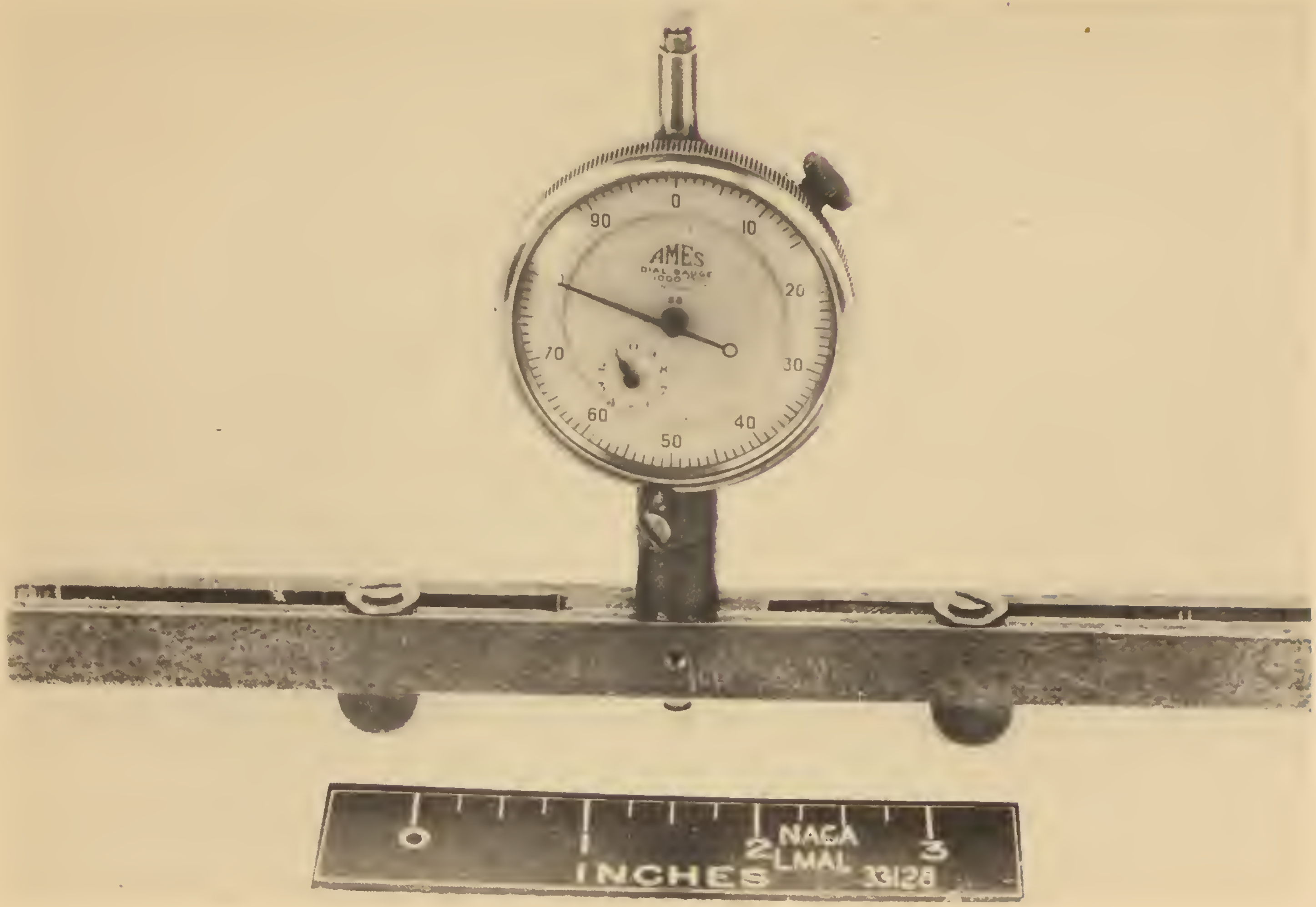

Figure 3.- Curvature gage used in measuring surface waviness.

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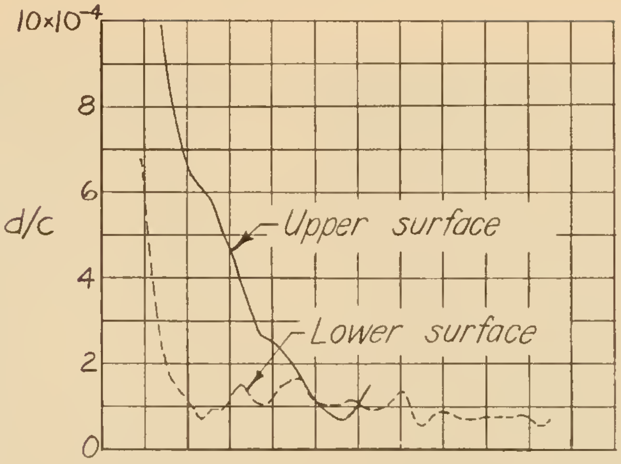
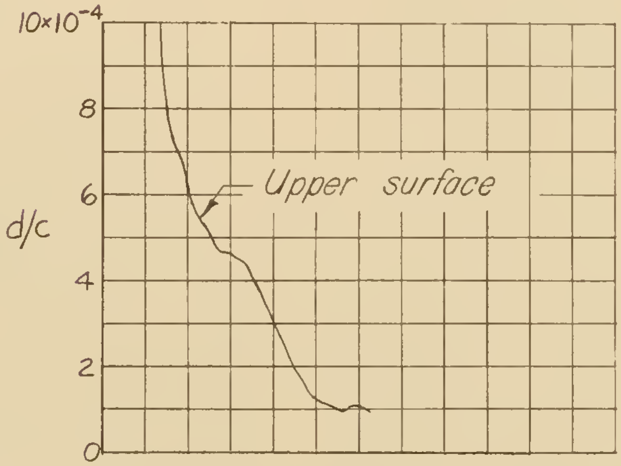
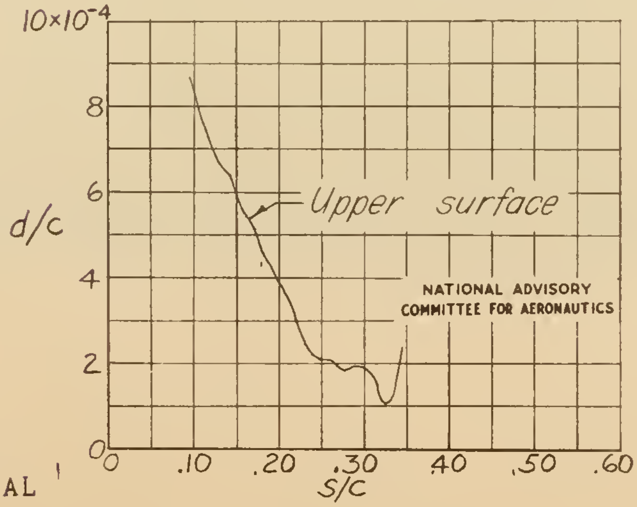


(a) At 63 percent semispan on right wing.



(b) At 63 percent semispan on left wing.



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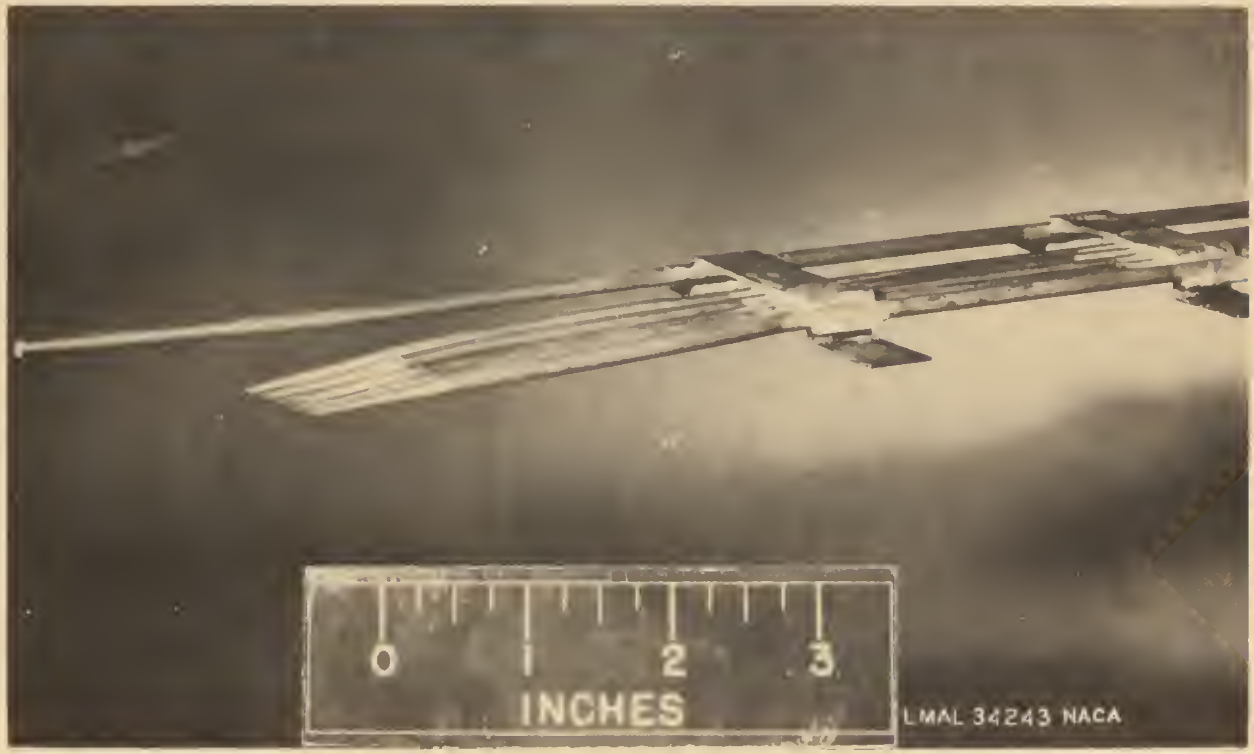
(c) At 25 percent semispan on left wing.

Figure 4.- Surface-waviness index of smooth surfaces of sections on right and left wings of P-47D airplane.



(a) Six-tube rack.

Figure 5.- Boundary-layer racks.

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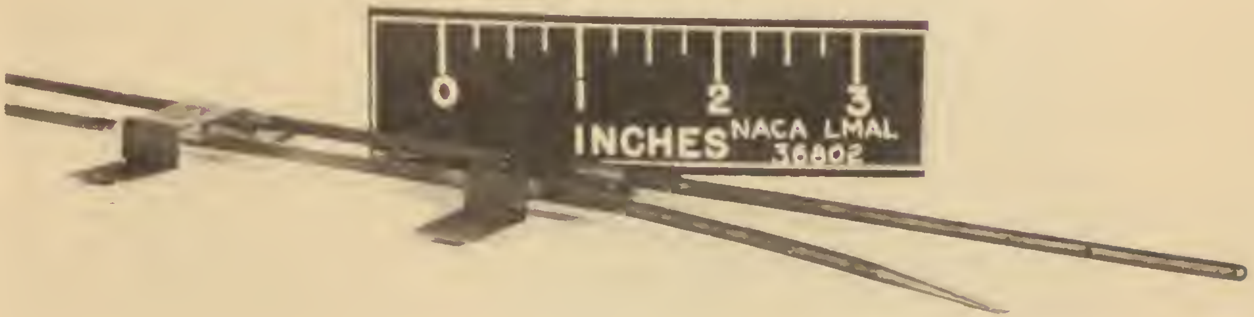

(b) Two-tube rack.

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Figure 5.- Concluded.

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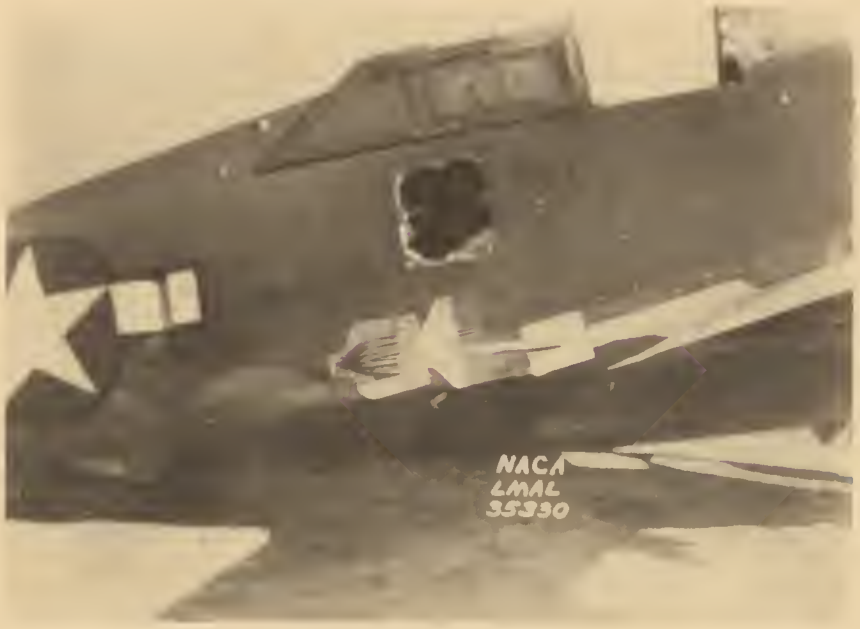

Figure 6.- Wake-survey rake mounted on wing of P-47D airplane.

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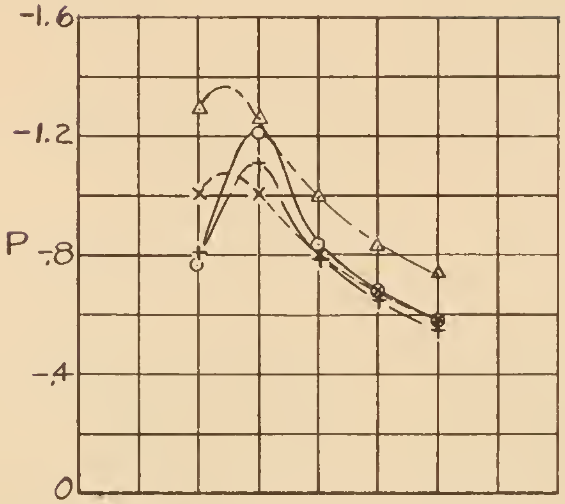
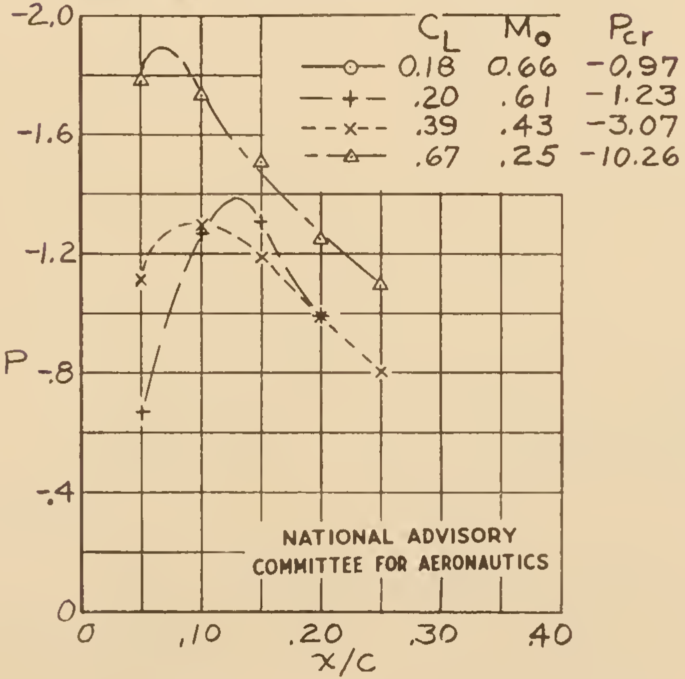
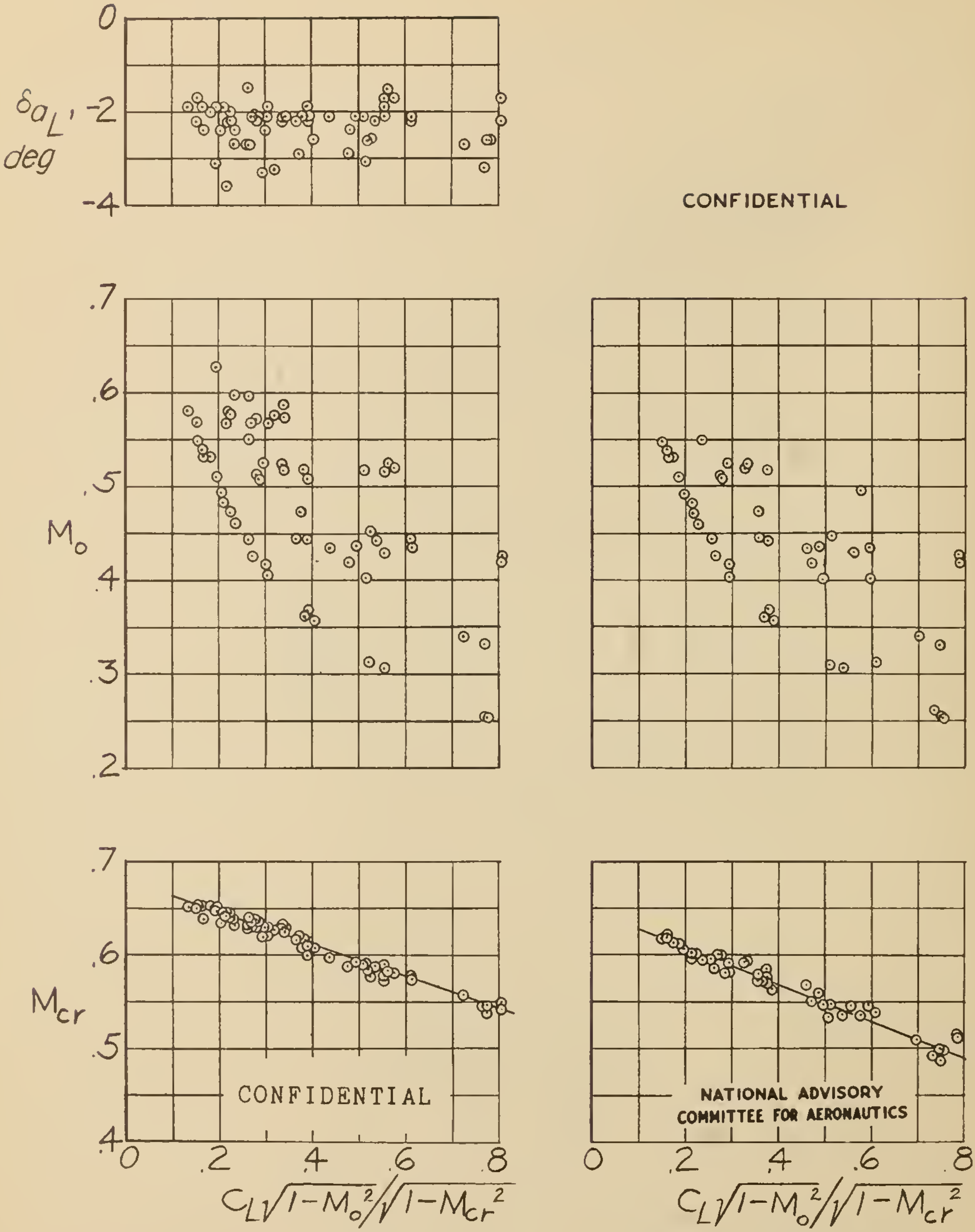


(a) At 63 percent semispan.



(b) At 25 percent semispan (in slipstream).

Figure 7.- Typical pressure distributions over upper surface of sections on left wing of P-47D airplane.



(a) At 63 percent semispan.

(b) At 25 percent semispan.

Figure 8.- Critical Mach number derived from subcritical pressure measurements on sections of left wing of P-47D airplane. Flight Mach number of pressure-distribution tests and deflection of left aileron are plotted above $M_{cr}$-curves.

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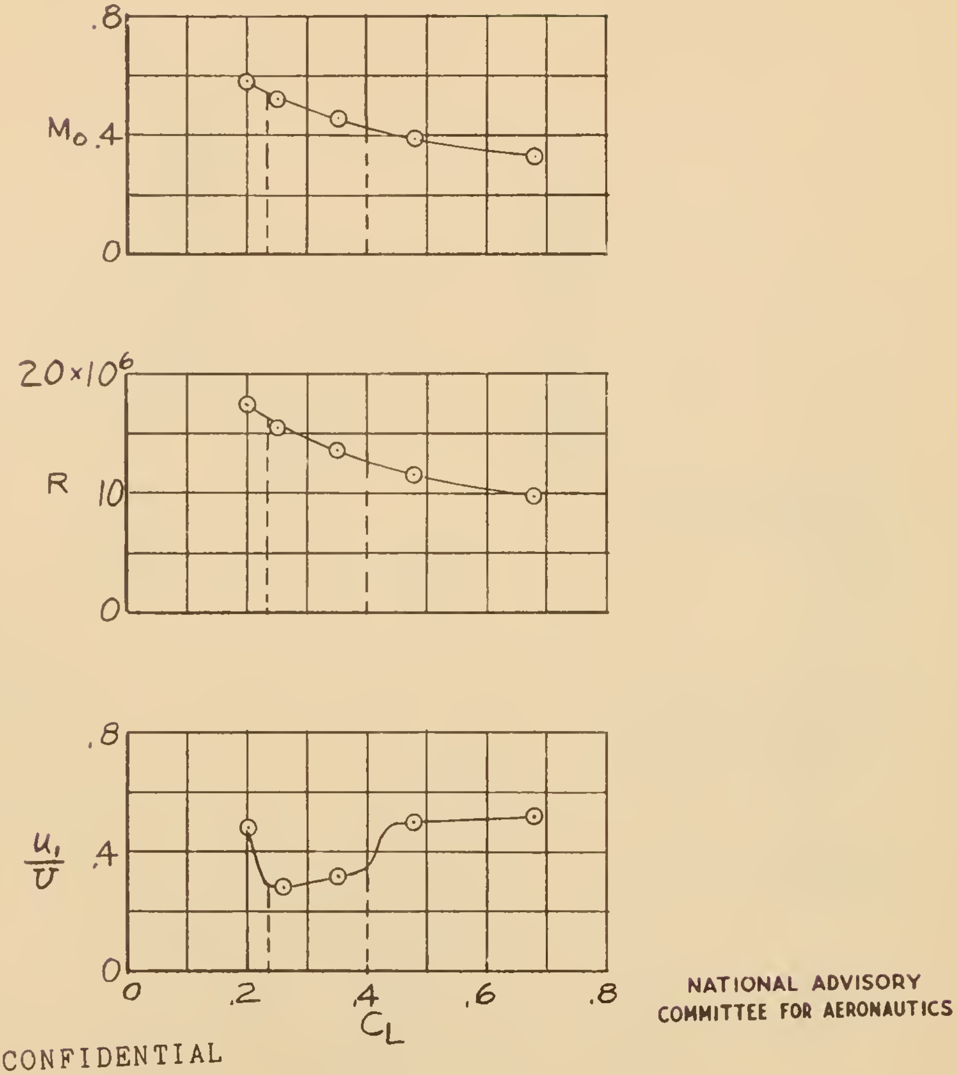

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Figure 9.- Method of determining lift coefficient, Reynolds number, and Mach number corresponding to transition at a given chordwise position. (Example shown is for 15 percent chord. Effective-pressure center of total-pressure tube at 0.01 in. above surface.)

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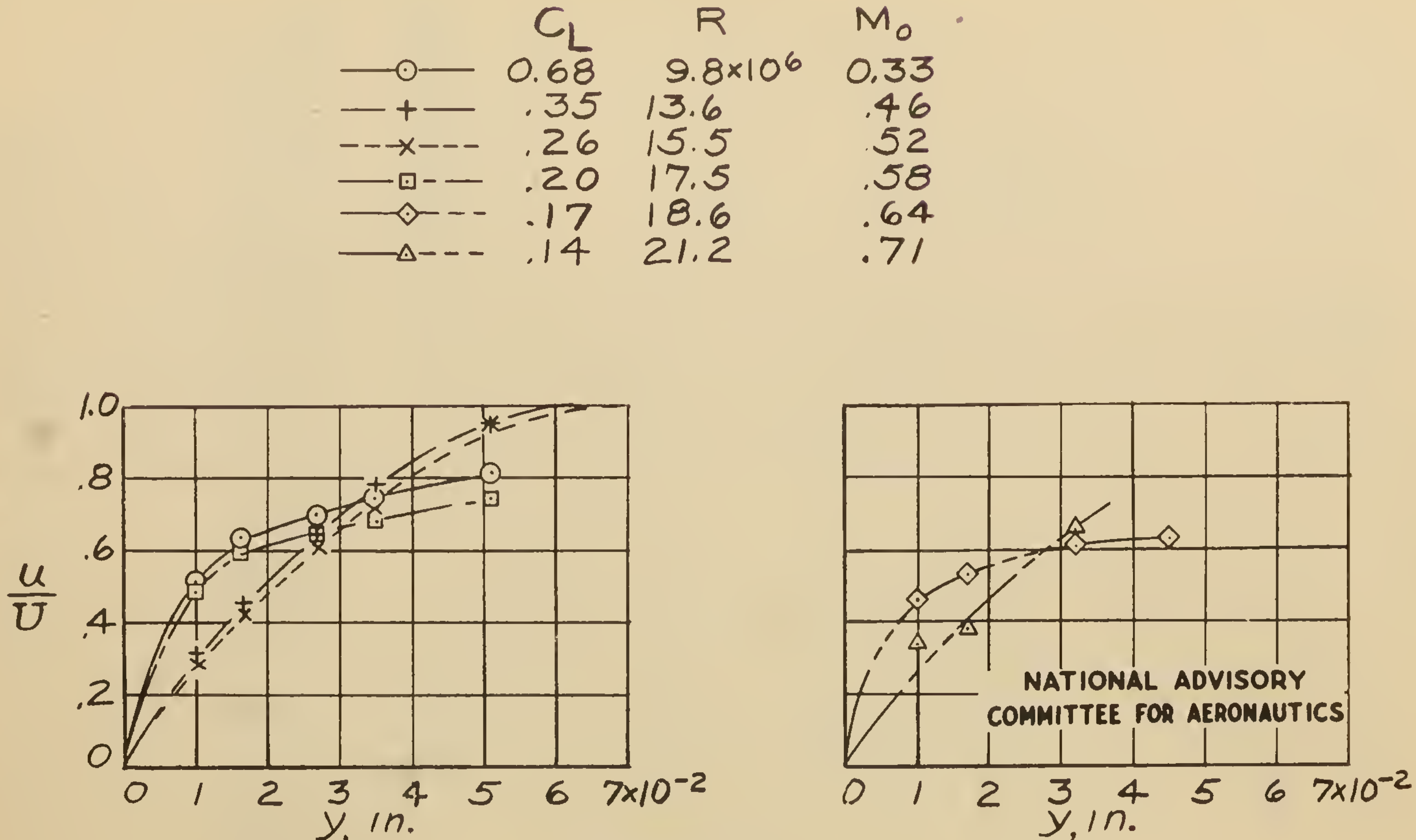

(a) x/c = 0.15.

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(b) x/c = 0.20.

Figure 10.- Velocity profiles in boundary layer on smooth upper surface of section of P-47D airplane wing in slipstream.

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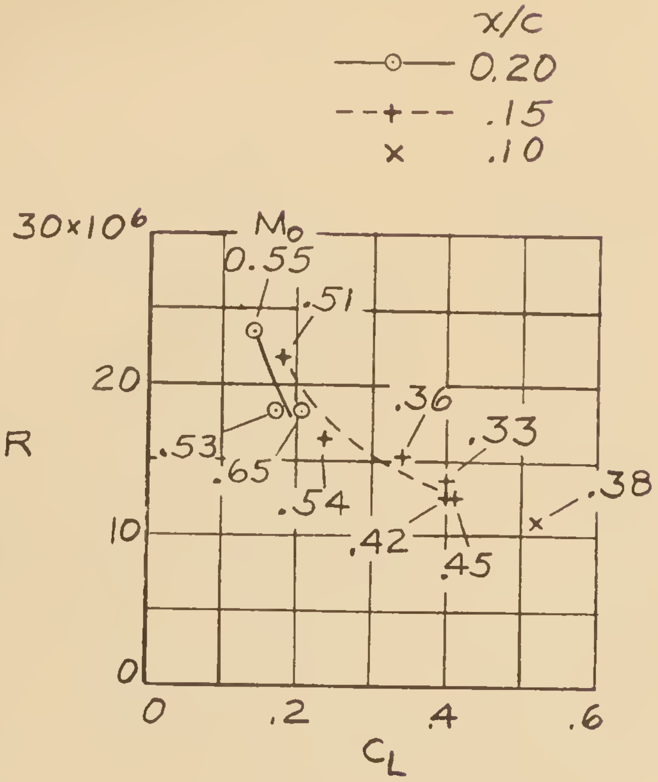

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Figure 11.- Relation of lift coefficient, Reynolds number, and Mach number for transition at three positions on smooth upper surface of section of wing on P-47D airplane in slipstream.

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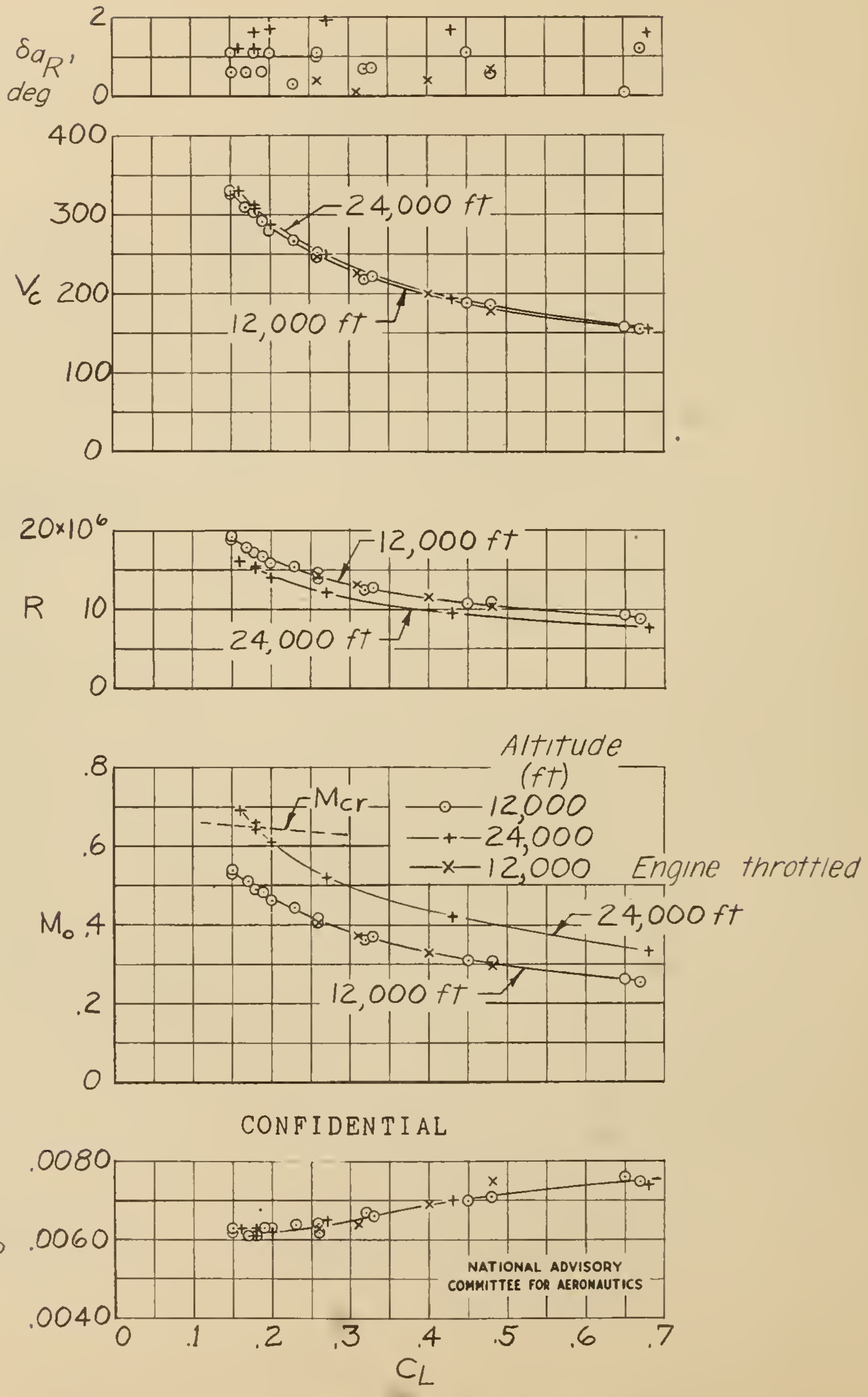

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Figure 12.- Profile-drag coefficient of smooth section on right wing of P-47D airplane as obtained in straight flight. Mach number, Reynolds number, calibrated airspeed, and deflection of right aileron are plotted above $c_{d_o}$-curve. $M_{cr}$-curve is from results of left-wing tests.

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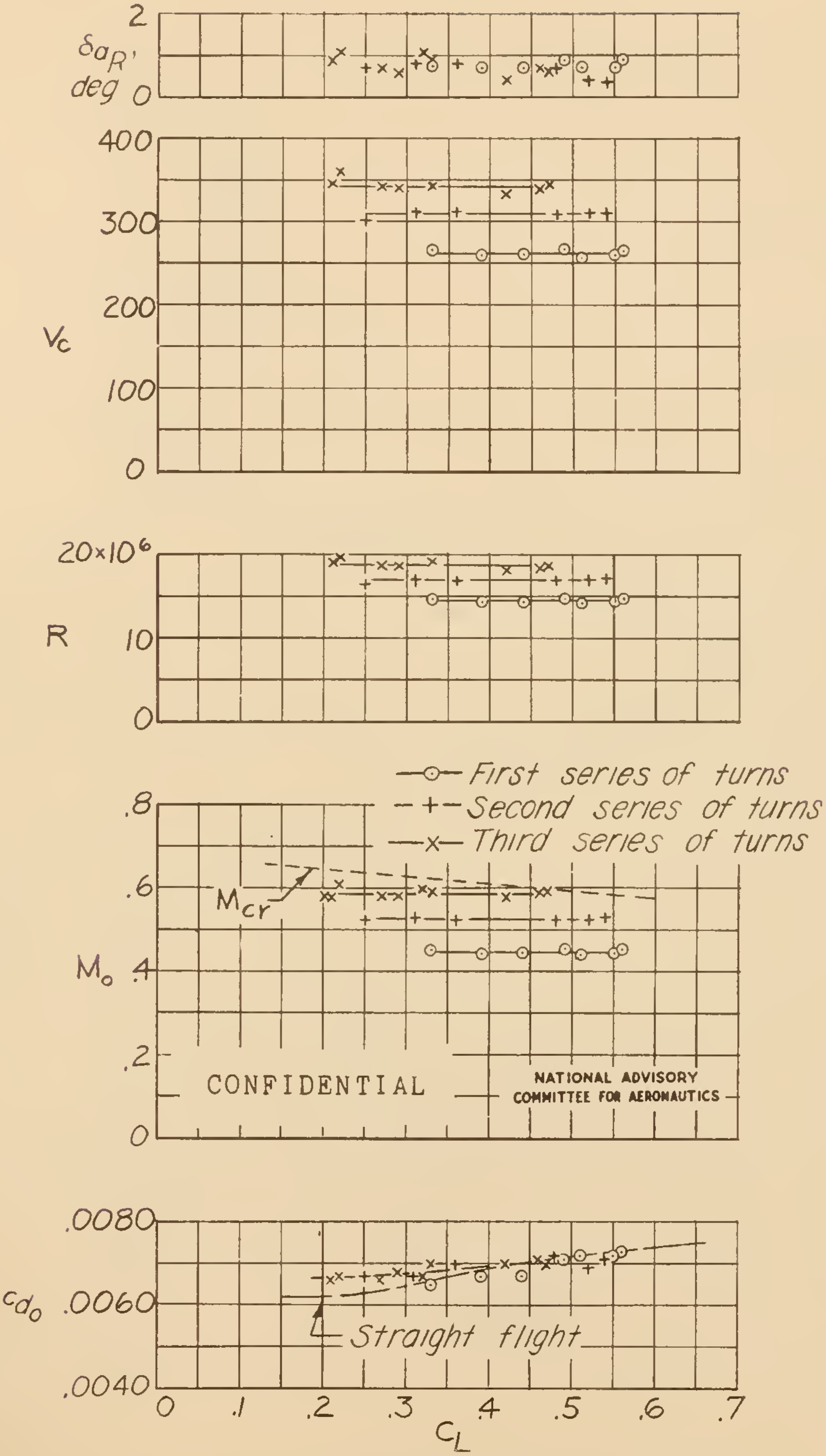

Figure 13.- Profile-drag coefficient of smooth section of right wing on P-47D airplane as obtained in turns. Mach number, Reynolds number, calibrated airspeed, and deflection of right aileron are plotted above $c_{d_o}$-curves. $M_{cr}$-curve is from results of left-wing tests.

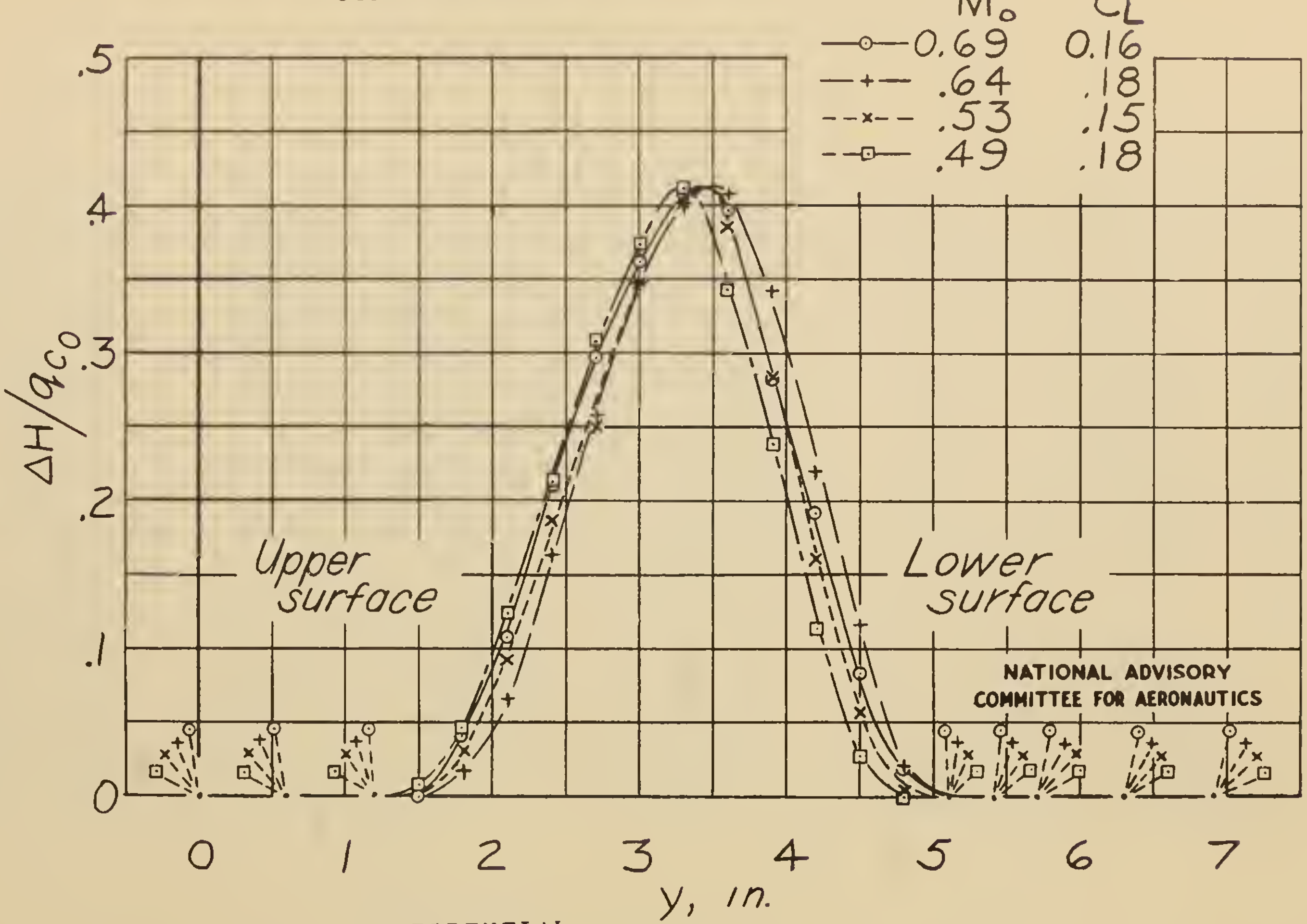

Figure 14.- Wake profiles of smooth section of right wing on P-47D airplane as obtained in straight flight. (Position y = 0 corresponds to top tube of wake-survey rake.)

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